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RESEARCH MEMORANDUM

EFFECT OF MACH NUMBER ON BOUNDARY-LAYER TRANSITION IN THE PRESENCE OF PRESSURE RISE AND SURFACE ROUGHNESS ON AN OGIVE-CYLINDER BODY WITH COLD WALL CONDITIONS

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NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

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SUMMARY

The effect of Mach number variation from 1.8 to 7.4 on boundary-layer transition was investigated on a slender fin-stabilized ogive-cylinder body in free flight at a constant length Reynolds number of 13.8 million. The wall to free-stream temperature ratio was constant at a value of 1.0 below Mach number 4.5 and at a value of 1.8 above Mach number 4.5. Results of the test showed that increasing Mach number had a very favorable effect of increasing the extent of the laminar boundary layer for a given surface roughness. The transition data, when plotted as a function of a factor indicative of heat transfer, showed that heat transfer was possibly responsible for a good deal of the increase in transition Reynolds number with Mach number.

Transition was found to occur farther forward on the sheltered side of the body than on the windward side for angles of attack as low as 0.4° and for all Mach numbers. The pressure rise along sheltered-side streamlines was examined and it was found that the pressure-rise coefficient at the transition point, showed no variation with Mach number. Data from other sources for different test conditions, when reduced to values of pressure-rise coefficient, were also found to correlate well with that of the present investigation with the exception of data at low subsonic Mach numbers. These present results also show that Mach number, surface roughness, pressure rise, and length Reynolds number all affected boundary-layer transition in the region of theoretical infinite laminar stability to small two-dimensional disturbances as calculated for a flat plate with zero pressure gradient.

INTRODUCTION

Aerodynamic heating resulting from friction is one of the major problems faced by designers of supersonic vehicles and, as is well known, is

very different for laminar and turbulent boundary layers. Therefore, it is a primary concern of the designer to know the extent to which the boundary-layer flow is laminar or turbulent so that cooling requirement calculations for the vehicle can be made. If the operating conditions of the vehicle are such that heating is of little concern, knowledge of the extent of laminar flow can nevertheless be important in determining the efficiency of flight.

The theoretical work of Lees and Lin (ref. 1) and Van Driest (ref. 2) predicts that for small two-dimensional disturbances the stability and extent of laminar flow will be increased by heat flow from the boundary layer to the body. This same analysis shows that on a flat plate with zero pressure gradient for Mach numbers between 1 and 9, if the heat transfer is sufficient, the laminar boundary layer will be stabilized for all values of Reynolds number. Investigations such as those of Scherrer (ref. 3) and Czarnecki (ref. 4) have confirmed experimentally the beneficial effect of heat transfer to the body. The experiments of reference 5 in the predicted regime of infinite laminar stability showed that transition will occur in this regime on roughened surfaces or in the presence of adverse pressure gradient. However, the amount of data collected in this regime, references 5, 6, and 7, is thus far rather small, and to the author's knowledge, is limited to Mach numbers below 3.7.

The present investigation was initiated at the NACA Ames Laboratory primarily to determine how Mach number affects transition within the predicted regime of infinite laminar stability. Previous wind-tunnel data, reference 8, had shown a decrease in transition Reynolds number with rising Mach number for the condition of small heat transfer.[1] A limited number of observations had been made also of the effect of Mach number on transition Reynolds number for the condition of constant wall temperature near stream static temperature. These observations, from the supersonic free-flight wind tunnel and other sources, showed a strong stabilizing influence on the laminar boundary layer of increasing Mach number. A purpose of the present test, then, was to investigate systematically the effect of Mach number on transition Reynolds number for the condition of constant, low wall to free-stream temperature ratio. In addition, the results obtained provide information on the effect of surface roughness, since roughness was varied to position transition in the field of view on the model and therefore became a necessary part of the investigation. As was the case in reference 5, it was observed that pressure rise was also affecting transition position when transition was not controlled by roughness. This effect became a part of the investigation and is considered along with the effects of Mach number and surface roughness.

[1]Potter (ref. 9) had suggested that the observed effect of Mach number on transition in the Naval Ordnance Laboratory wind tunnels was influenced by other factors in addition to Mach number.

SYMBOLS

d	body diameter, in.
h	height of roughness, in.
l	body length, in.
l_n	ogive nose length, in.
M	Mach number
p	local static pressure, lb/sq ft
p_o	free-stream static pressure, lb/sq ft
$\frac{\Delta p}{q_o}$, $\frac{\Delta p}{p_o}$	pressure-rise coefficients (difference between the pressure coefficient at a particular body station and the minimum pressure coefficient along a streamline)
$\left(\frac{\Delta p}{q_o}\right)_{crit}$, $\left(\frac{\Delta p}{p_o}\right)_{crit}$	critical pressure-rise coefficients (the pressure-rise coefficient above which transition due to pressure rise will occur)
q_o	free-stream dynamic pressure, lb/sq ft
R_l	Reynolds number based on free-stream properties and body length
R_x	Reynolds number based on free-stream properties and distance x
R_{crit}	critical free-stream Reynolds number above which the effects of small disturbances to the boundary layer are amplified
R_T	instantaneous transition Reynolds number based on free-stream properties and length of run of the laminar boundary layer
$(R_T)_{av}$	arithmetic average of instantaneous transition Reynolds numbers
T_o	free-stream static temperature, ^{o}R
T_r	boundary-layer recovery temperature, ^{o}R
T_w	temperature of model surface, ^{o}R

x	axial distance from body nose, in.
α	angle of attack, deg
θ	meridian angle of model measured from the windward side of the body, deg

TEST DESCRIPTION

In this investigation in the Ames supersonic free-flight wind tunnel models were launched from a caliber 50 smooth-bore test gun at velocities from 2000 to 5500 feet per second. Some models were fired through still air and others were fired upstream through the Mach number 2 air stream of the wind tunnel. The models, shown in figures 1 and 2, were fin-stabilized, ogive-cylinder bodies of fineness ratio 30 and were made of 75 ST-6 aluminum alloy. They were launched from the gun with the aid of plastic sabots shown in figure 2. The models in flight passed through a group of shadowgraph stations located from 40 to 55 feet from the gun muzzle, providing four shadowgraph pictures in the horizontal plane and three in the vertical plane. The reader will find a more detailed description of the facility and techniques in reference 10.

For the most part the model surfaces tested were continuous screw threads of desired depth starting at approximately 0.05 inch from the tip and extending to the stabilizing fins. This type of roughness was selected because it could be controlled very well and could be repeated from one model to another. Out of a total of 26 models, 4 were finished with a controlled sandblast operation which produced a three-dimensional-type surface roughness. The sand driven against the model surface by the blast of air caused the metal to flow up and form minute craters and in some cases the sand was actually imbedded in the surface. The surface was covered by an average of 3500 craters per square inch and the height above the free surface varied from 0 to 0.001 inch. The first 0.05 inch of each model tip was hand-polished to insure that it would be smooth, symmetrical, and the same for all models. The model surface conditions were examined very carefully and recorded by use of a metallurgical microscope up to a magnification of 550x. Typical photomicrographs of a nose-tip profile, screw-thread profile, and line-shadow profile[2] are shown in figure 3. These types of photographs were used to record the model tip and screw-thread condition.

The tests were conducted in a range of Mach numbers from 1.8 to 7.4 at a nominally constant Reynolds number per inch of 2.3 million. The

[2]The line-shadow profile in this case was a shadow of a fine straight wire cast obliquely on a surface for the purpose of examining the profile of that surface. The basic principle was devised by Schmalz, reference 11.

pressure in the test chamber was varied from 0.4 to 2 atmospheres absolute to maintain a constant test Reynolds number. In the range of Mach numbers from 1.8 to 4.5 for which the air in the test section was still (hereinafter designated air-off), the wall to free-stream temperature ratio was 1.0. At test Mach numbers from 4.5 to 7.4, models were launched through the wind-tunnel air stream (hereinafter designated air-on). The wall to free-stream temperature ratio in this case was 1.8. The relationship of these temperature ratios to those theoretically required for infinite laminar stability to small two-dimensional disturbances on a flat plate with zero pressure gradient is shown in figure 4. A calculation was made, which was similar to that described in the appendix of reference 5, to see what the wall to free-stream temperature ratio was at the very tip of the model. This calculation showed that the first few hundredths of an inch of the model tip had a wall temperature ratio of from 1.0 to 2.9. Since this temperature rise was confined to the very tip of the model, it was believed to have a negligible effect on transition.

The stream turbulence in the test chamber for air-off testing was zero since there was no movement of the air in this case. Stream turbulence was present in the case of the air-on testing, but no measurements of its magnitude are available. The effect of this turbulence on the data will be discussed under "Results and Discussion."

DATA REDUCTION

Transition to turbulent flow was determined from the shadowgraph pictures by the appearance of eddies in the boundary layer which obliterate the diffraction line associated with a thin laminar boundary layer and by the appearance of Mach lines in the flow field adjacent to the turbulent boundary layer. An example shadowgraph record with transition position located as explained is shown in figure 5. Evidence that transition position as determined optically agrees with transition position determined by such means as a probe has been shown by a number of investigators (see, e.g., ref. 12). Each shadowgraph picture provided a position of transition on the windward and sheltered sides of the body and an angle of attack. A total of 14 observations of the transition location were therefore made for each model flight as well as a record of the model pitching history.

Transition Induced by Roughness

When surface roughness was sufficient to control transition position, the 14 observations for a single model flight showed unsteady movements of the transition point over a range of Reynolds numbers of from 1 to 3 million in extent and were reduced to a single value of transition Reynolds number in the following manner.

The percentage of observations in which a given body station was found to be turbulent was noted and plotted as in figure 6. At the body station where the Reynolds number was 6 million, 5 out of a total of 14 observations showed turbulent flow, indicating that this body station was turbulent approximately 36 percent of the time. The boundary layer was fully laminar to a Reynolds number of 4 million, transitional from 4 to 8 million, and fully turbulent beyond 8 million. For the purpose of comparing in a simple manner the transition location of the separate models, a single value of transition Reynolds number was assigned to each. The station at which the boundary layer was turbulent 50 percent of the time was selected as the location of transition.

Transition Induced by Angle of Attack

When surface roughness was not sufficient to control transition in itself, angle of attack did influence the position and the data were reduced to determine the value of transition Reynolds number at zero angle of attack in the following manner. The transition observations were separated into windward- and sheltered-side data and plotted against angle of attack as in figure 7. Representative data presented in this figure show more of a spread in windward- and sheltered-side transition as the angle of attack was increased. The angles of attack used here were the resultant angles with respect to the wind direction and were determined from the shadowgraph pictures in the horizontal and vertical planes. As can be seen from the figure, the transition Reynolds number for $\alpha = 0^{\circ}$ can be well defined by extrapolation of observations at $\alpha \neq 0^{\circ}$, for both the windward and sheltered sides of the body.

RESULTS AND DISCUSSION

The effects of Mach number, surface roughness, and pressure rise on transition Reynolds number which were observed in this test are presented and discussed in the following sections.

Effects of Mach Number and Surface Roughness

The experimental results of the effect of Mach number on transition Reynolds number for several surfaces of controlled roughness height are presented in figure 8. The Mach number range extends from 1.8 to 7.4 and the Reynolds number per inch was maintained nominally constant at 2.3×10^6. A large variation in transition Reynolds number was observed as the Mach number was increased at constant wall to free-stream temperature ratio, as can be seen from an examination of the result obtained for the models

with 0.0004-inch-depth screw-thread surfaces. A change in Mach number from 1.9 to 3.4 resulted in a change in transition Reynolds number from 1 million to 12 million.

Beyond a Mach number of 3.4 it was necessary to increase the roughness height in order that transition would be moved forward on the body into the region of observation. A 0.0007-inch thread was tried and found to produce the result shown. As the Mach number was increased from 4.5 to 5.2, there was a marked rearward movement of transition similar to that observed with the 0.0004-inch thread at the lower Mach numbers. Above a Mach number of 5.2 the curve is dashed to indicate uncertain fairing since the transition point was off the body at $M = 6.8$ and was not clearly defined at $M = 6.4$. These data indicate a possibility that transition Reynolds number may not increase indefinitely with Mach number. More experimental work is necessary to clear up this point. Below a Mach number of 4.5 transition occurred in the region of the ogive-nose of the body and the slope of the curve changed as shown. Mach number still had an effect on transition but it was much less than when transition occurred back on the cylindrical part of the body. A value for R_T was obtained from reference 5 for an intermediate value of surface roughness, a 0.0005-inch screw thread, and was found to fall as would be expected at an intermediate location between the curve for the 0.0004-inch thread and the curve for the 0.0007-inch thread.

For the Mach numbers above 5 it was evident that a rougher (deeper) screw thread would be necessary to move transition into the field of observation on the model. The first to be tried was a 0.0010-inch thread at a Mach number of 5 which gave a value of R_T beyond the limit of observation. The second attempt was made with a 0.0020-inch thread at a Mach number of 6.5 which gave a value of R_T on the ogive. The desired depth thread was evidently somewhere between. Thus a 0.0015-inch thread was tried which produced the result shown in the figure. At Mach numbers above 5, the slope of the curve is somewhat less than with the finer screw threads at lower Mach numbers, but the effect of increasing Mach number is still large. It is interesting to note from the curves for the various screw threads that as Mach number is increased from 2.7 to 5.3 it was necessary to increase the screw-thread depth considerably, from 0.0004 inch to 0.0015 inch, in order to keep transition at a constant value of R_T of 7.4 million.

In the course of the investigation it was wondered if the effect of M on R_T being observed could possibly be influenced by the type of surface roughness used to control transition. For this reason a three-dimensional type roughness (sandblasted surface described earlier) was tested in the Mach number range shown. The slope of the curve is similar to the 0.0015-inch-thread curve, thus indicating that the effect of Mach number on R_T being observed was not confined to the screw-thread surfaces. It is also interesting to note that although the sandblasted surface had projections of less height (in the order of 0.001 inch maximum)

than did the 0.0015-inch thread, it gave a lower value of R_T for the same Mach number than did the screw thread. This would indicate that a three-dimensional random type of surface roughness is more damaging to a laminar boundary layer than a two-dimensional regular type of surface roughness of about the same height.

In figure 8, data are presented for two basically different conditions of testing, that is, for air off and for air on. Connected with the air-off conditions are zero air-stream turbulence and a temperature ratio, T_W/T_O, of 1.0. In the air-on tests the air stream is turbulent and the temperature ratio is 1.8. Let us first consider air-stream turbulence. In the case of the air-off testing there was no turbulence. In the air-on case, however, there was without doubt some turbulence present in the air stream and one would expect this to affect transition, if at all, in such a manner as to produce earlier transition. It was believed that, while the turbulence for the tunnel was not known, the value of the fluctuating velocity of the air stream would be small compared to the resultant velocity of the model through the air. For this reason and because the transition data of figure 8 (for the 0.0007-inch screw thread) did not show a decrease in R_T from the air-off to the air-on conditions, it can be assumed for all practical purposes that transition was not seriously affected by the air-stream turbulence. Furthermore, increasing the temperature ratio from 1.0 air off to 1.8 air on would be expected to cause transition to occur at a lower value of Reynolds number. In a change of test conditions from air off to air on, both air-stream turbulence and increased temperature ratio act in such a manner as to cause transition to occur earlier on the body and since no such case of earlier transition was observed in the data of figure 8, it was concluded that the effect of these two variables was small and overshadowed by other effects in the present tests.

For the transition results discussed above, heat transfer from the boundary layer to the model increased with increasing Mach number roughly in proportion to the temperature-difference ratios shown in figure 9. When the transition data of figure 8 are plotted against the temperature-difference ratios (fig. 10), the result shows a favorable effect of cooling on increasing transition Reynolds number. This result agrees with earlier findings of other investigators (see, e.g., refs. 4 and 13) on the effect of cooling on R_T at constant Mach number. The similarity observed in these results suggests that increasing boundary-layer cooling is one factor (perhaps the principle factor) causing the increase in R_T with increasing Mach number. As roughness is increased, an increased amount of cooling is required, but if sufficient cooling is applied, transition is forestalled on even the roughest models tested. Other factors known to influence transition and that vary with Mach number are pressure gradient, boundary-layer thickness, and boundary-layer profile. To what extent these factors contribute to the favorable effect of increasing Mach number on transition Reynolds number can not be determined from the present tests.

Effects of Pressure Rise

The influence of angle of attack on transition was present on several of the models tested. The location of transition on the windward and sheltered sides of the body was different and the variation of transition on the sheltered side with α was interpreted, as in reference 5, to be a result of pressure rise due to angle of attack. The authors of reference 5 were able to define the variation of transition Reynolds number with angle of attack to a much better degree than in the present investigation since all of their data were collected at one Mach number. The scope of the present test was such as to obtain data over a wide range of Mach numbers and thus the variation of R_T with α was not as well defined for any one Mach number. However, an increased sensitivity of the transition location to angle of attack for angles less than approximately 1^o did appear to be present at the higher Mach numbers of the present test, as indicated in figure 11. This figure shows data from the present test in the Mach number range from 5 to 7 and also shows data of reference 5 reproduced for comparison. The data of figure 11 show the increased sensitivity to angle of attack in that the minimum angle of attack for differences between windward and sheltered transition location was reduced in the present investigation. This increased sensitivity to angle of attack was deemed to be of interest and importance because it represented an adverse effect of increasing Mach number. It was this observation which led to the attempted correlation of critical-pressure-rise coefficient with Mach number.

The pressure-rise coefficients, $\Delta p/q_o$, associated with these observations were computed by a method similar to that described in reference 5 and is reviewed briefly here for the convenience of the reader. The axial pressures were obtained from reference 14 and the crossflow pressures were obtained by use of slender-body theory as in reference 15. The streamline paths used were obtained from the method of reference 16 in which it was assumed that the incompressible distribution of crossflow velocity around the cylinder applies and that the axial velocity component is the same as for $\alpha = 0^o$. The calculation was made for several streamlines for angles of attack of 1^o, 2^o, and 3^o and the result for $\alpha = 2^o$, $M = 6.8$ is shown in figure 12. One would expect transition to first occur in the streamline having the maximum pressure rise for that body station.[3] Therefore, these values of $\Delta p/q_o$ were used for corresponding transition locations. For example, if transition occurred at a body station of $X = 2.2$ inches, the corresponding value of pressure rise would be 0.0061, the maximum value at that body station. This maximum value of $\Delta p/q_o$ for the example occurs on the streamline intersecting the nose-cylinder juncture at $\theta = 107^o$ or 17^o above the side of the body.

[3]For information on the variation of sheltered-side transition with meridian angle, see reference 5.

The sheltered-side transition data were reduced to $\Delta p/q_o$ by this procedure and were plotted versus Mach number as shown in figure 13.

The data of the present test plotted in figure 13 show no systematic correlation of R_T with $\Delta p/q_o$. This result is not consistent with figure 22(a) of reference 5 which showed, at $M = 3.5$, a dependence of pressure-rise coefficient on transition Reynolds number. The reason for this disagreement is not clear. It may be due to the small number of observations at each Mach number and the small Reynolds number range[4] of the present data which would tend to emphasize the effects of experimental scatter and unsteadiness of the transition point. Further experimental work is evidently required, then, to determine the dependence of pressure-rise coefficient on R_T. What is evident from the present correlation is that Mach number has an important influence on the pressure-rise coefficient $\Delta p/q_o$ and, therefore, that the correlation of data for various Mach numbers attempted in reference 5 on the basis of Reynolds number alone could not succeed.

After the trend of $\Delta p/q_o$ with Mach number was observed in this figure, it was believed that possibly the parameter $\Delta p/p_o$ would be a better one to use since, in the former quantity, $\Delta p/q_o$, q_o has a dependency on Mach number. When the transition data were transposed to values of $\Delta p/p_o$ and plotted in figure 14, the correlation showed no dependence on Mach number within the scatter of the data. The figure shows the values of $\Delta p/p_o$ to range from 0.135 to 0.215 with a mean value of 0.175. Indications are that pressure-rise transition will occur at approximately this mean value for all of the Mach numbers investigated. The data presented include a change in temperature ratio, T_W/T_O, of from 1.0 at $M = 3.5$ to 1.8 at $M = 5$ and 6.8. No effect of this change in temperature ratio was observed.

To check the effect of pressure-rise transition without the influence of crossflow pressure rise due to angle of attack, a polished cone-cylinder body with an axial pressure-rise sufficient to cause transition at $\alpha = 0$ was launched at $M = 5$. Transition due to pressure rise at $\alpha = 0$ did occur, as would be predicted from figure 14, at the body station where $\Delta p/p_o$ was 0.17. The data point obtained from this test is included on the figure.

Data of other investigations were reduced to see if the correlation would hold true for the lower Mach numbers and for other configurations. The data were obtained on NACA airfoils (ref. 17), on a monoplane wing (ref. 18), on an ogive-cylinder and cone-cylinder (ref. 19), and on a cone-cylinder (ref. 9). Theoretical pressure distributions were used in

[4]The transition Reynolds number range of reference 5, figure 22(a), was from 4.5 million to 11 million; whereas in figure 13 of the present report the range is from 4 million to 7 million.

determining the pressure-rise coefficients in the cases where no experimental distributions were available. Theoretical pressure distributions were obtained from characteristics solutions for the ogive-cylinder shapes (ref. 14) and for the cone-cylinder shapes (ref. 20). Several sources were investigated to determine if in the experimental case the boundary-layer effects at the cone-cylinder juncture might alter the nature of the expansion at that point. The very little amount of pressure-distribution data that was found to include the region very near the cone-cylinder junction showed that the flow did not expand at the corner to as low a pressure as predicted by theory. This difference between the experimental and theoretical pressure coefficients at the corner was found to be close to an average of 20 percent. This value of 20 percent was used to reduce the theoretical pressure coefficients at the corner for the cone-cylinder bodies since it was the best information available. All of these data from other investigations except for the subsonic data correlate well and have very nearly the same value of $\Delta p/p_o$ for pressure-rise transition as the data of the present test.

When transition is predominately controlled by roughness, vibration, air-stream turbulence, etc., correlation with figure 14 should not be expected. In addition, all of the data of figure 14 were obtained for cases where the boundary-layer thickness development did not depart radically from that for a flat plate. Caution should be applied, therefore, in using this correlation on shapes when the boundary-layer thickness changes rapidly and extensively, as on boattailed bodies or flared bodies or when other conditions vary considerably from those of the present test.

It is interesting to note that as Mach number increases, the pressure rise to cause transition takes on increasing relative importance since surface roughness is becoming less important. That is to say, when long laminar runs are desired, smoothness is more important than pressure rise at the low Mach numbers, but as Mach number increases, surface smoothness becomes less important and pressure rise is of more concern.

Long Laminar Runs

Aside from the original plan of the investigation, since increasing Mach number showed such a favorable effect on increasing the length of laminar run, two models polished with fine emery paper were launched in an attempt to obtain a high value of transition Reynolds number. Both rounds were launched at $M = 7$, the first at the maximum length Reynolds number available, 36 million, and the second at 22.8 million. The static pressure in the wind tunnel was varied to obtain this change in length Reynolds number. The first round gave a value of 15 million for R_T. The second test made at the lower Reynolds number gave a value of 11.6 million for R_T. These laminar runs, while fairly long, were not as great as had been expected. However, the test conditions were such as to

produce a very thin laminar boundary layer, order of a few thousandths of an inch thick, and this imposes very stringent requirements on surface smoothness. For tests at larger scale and consequently with thicker boundary layers, the surface would not have to be as smooth as in the present case, and quite possibly higher values of transition Reynolds number might be attained.

Examination of the two results discussed in this section show that R_T increases with increasing length Reynolds number. This same trend was observed by Brinich, reference 21, and observed earlier by Witt in some data obtained in the NOL Pressurized Ballistics Range reported in reference 9. However, it is interesting to note that some data presented in reference 5 for models with a rough screw-thread surface showed a decrease in R_T as length Reynolds number was increased. This difference may be attributable to the difference in the degree of surface roughness.

CONCLUSIONS

Boundary-layer-transition data have been presented from free-flight tests of a slender body of revolution at Mach numbers from 1.8 to 7.4 and a constant length Reynolds number of 13.8 million. The wall to free-stream temperature ratio was constant at two levels, 1.0 and 1.8, and, therefore, the temperature difference ratio (which is indicative of heat transfer), varied with Mach number. Conclusions derived from this investigation are summarized below:

1. For the conditions described above, the laminar boundary layer extended to higher Reynolds numbers as Mach number was increased.

2. As the depth of surface roughness was increased, the Reynolds number of transition decreased, but the depth of roughness did not, in general, alter the influence of Mach number on transition.

3. The transition data, plotted against a boundary-layer cooling factor (which was a function of M), is in accord with earlier findings of other investigators on the effect of cooling on boundary-layer transition at constant Mach number. How much of the favorable effect of increasing Mach number can be attributed to boundary-layer cooling and how much to other factors such as pressure gradient, boundary-layer thickness, and boundary-layer profile could not be determined.

4. For the range of conditions of this investigation and others reported in the text, essentially the same value of pressure-rise coefficient caused transition at all supersonic Mach numbers.

5. For the slender body of revolution of the present test, it was observed that pressure rise became increasingly important in causing transition as Mach number was increased.

6. The highest transition Reynolds number attained in the present test at a Mach number of 7 and for a thin boundary layer was 15 million and was obtained on a model polished with fine emery paper.

7. The parameters of this investigation, namely, Mach number, surface roughness, pressure rise, and Reynolds number, were found to influence transition in the Lees-Van Driest region of predicted infinite laminar stability to small two-dimensional disturbances.

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National Advisory Committee For Aeronautics
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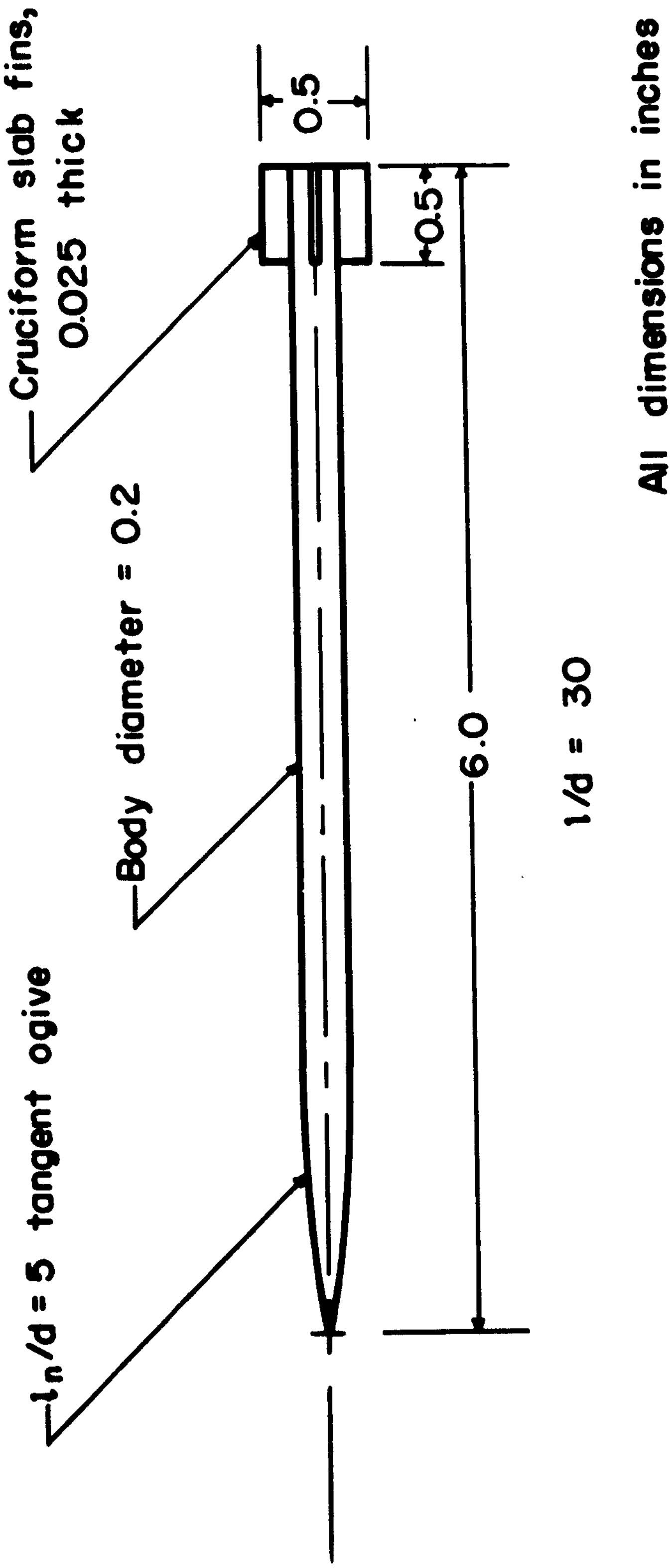
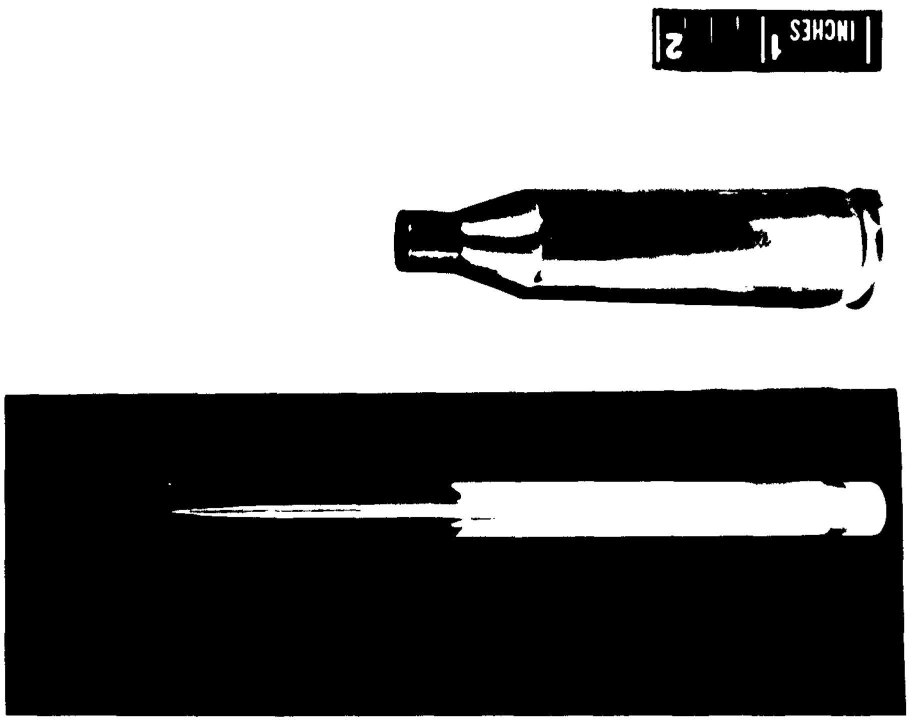


Figure 1.- Typical model dimensions.



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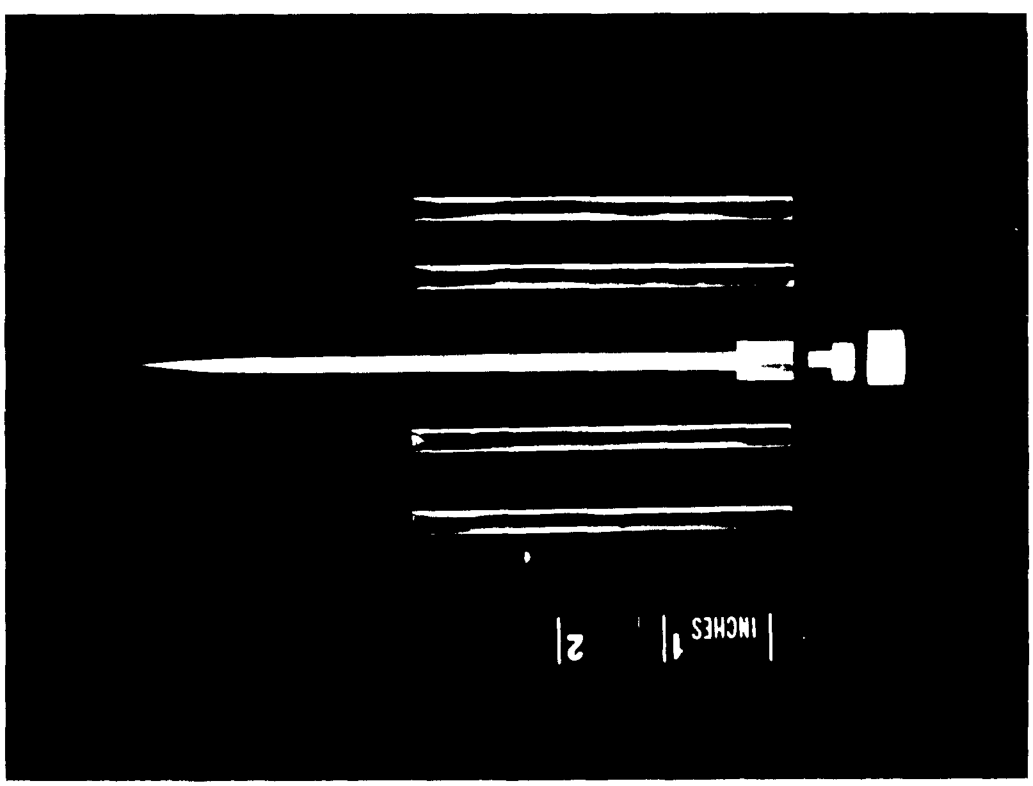
(b) Assembly.



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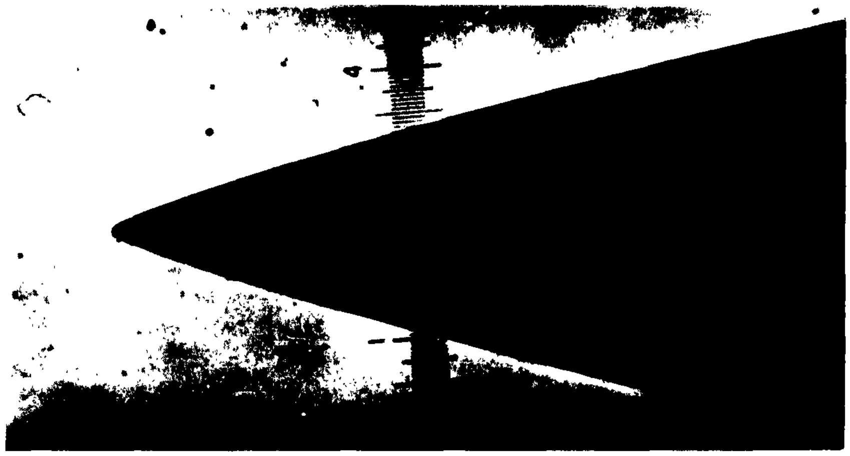
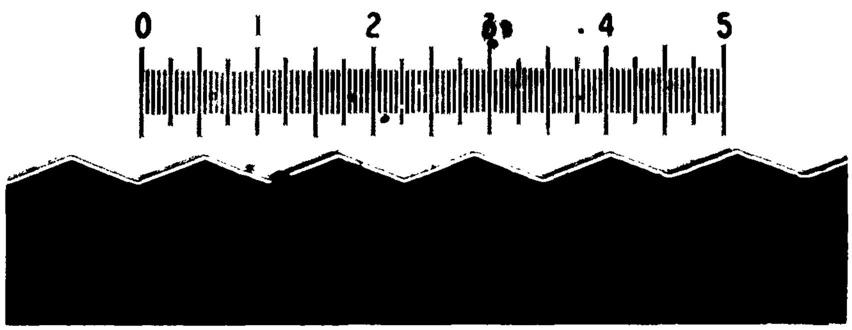
(a) Model and sabot parts.

Figure 2.- Model and sabot.



(a) Nose tip profile; 200 x.



(b) Screw-thread profile; h = 0.0015 inch; 100 x.

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(c) Line-shadow profile; h = 0.0007 inch; 550 x.

Figure 3.- Typical photomicrographs.

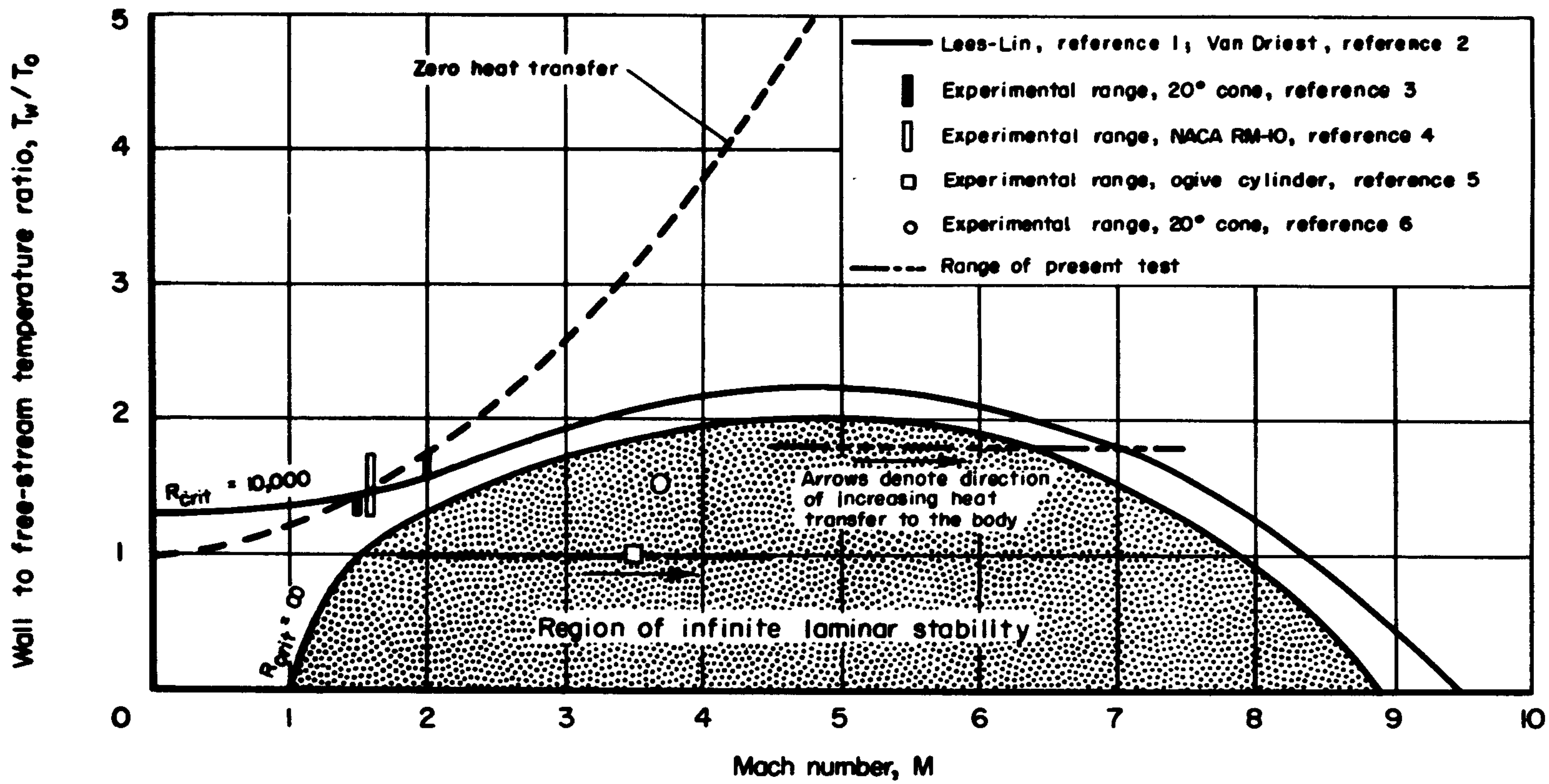
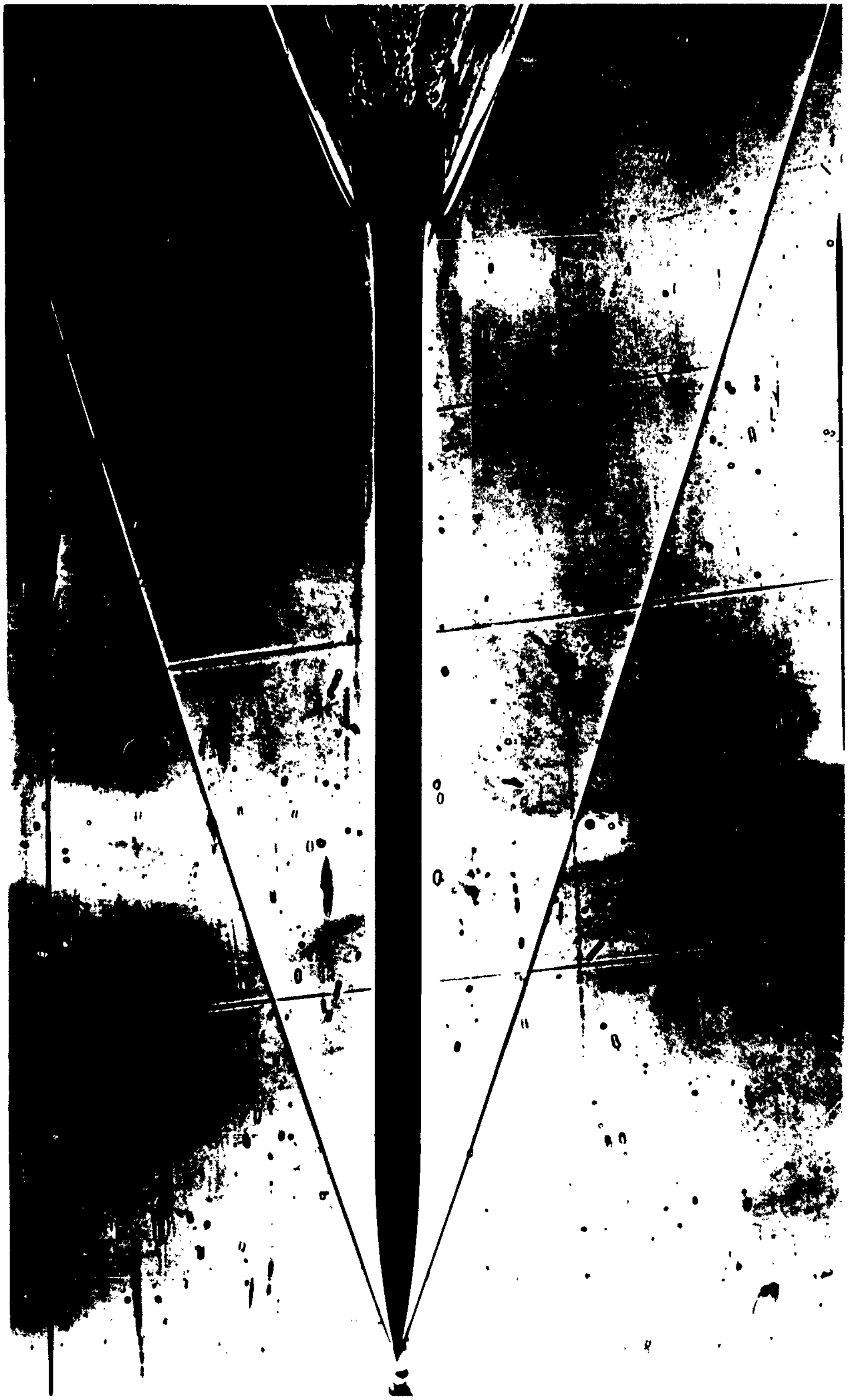


Figure 4.- The wall to free-stream temperature ratio of the present investigation as compared to the region of infinite laminar stability predicted for a flat plate with zero pressure gradient.

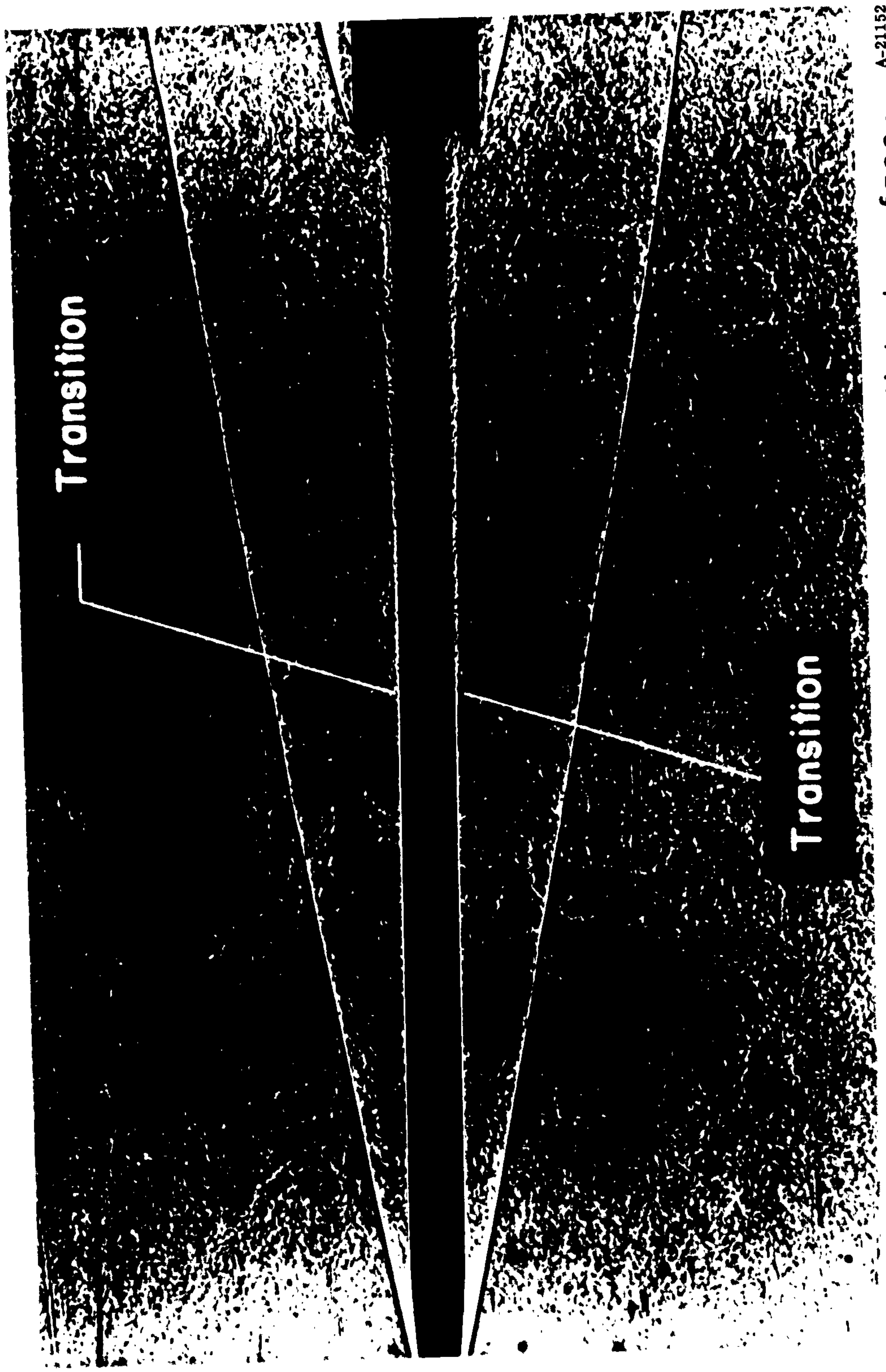


(a) Laminar boundary layer; emery-polished surface; $\alpha = 0.7°$; $M = 3.6$; $R_1 = 12$ million.

Figure 5.- Shadowgraphs of models in flight.

(b) Turbulent boundary layer; screw-thread surface; h = 0.0007 inch; $\alpha = 0.7°$; M = 3.4, R_l = 14 million.

Figure 5.- Continued.



(c) Transition in boundary layer; emery-polished surface; $\alpha = 0.2°$; $M = 6.8$; $R_l = 22.6$ million.

Figure 5.- Concluded.

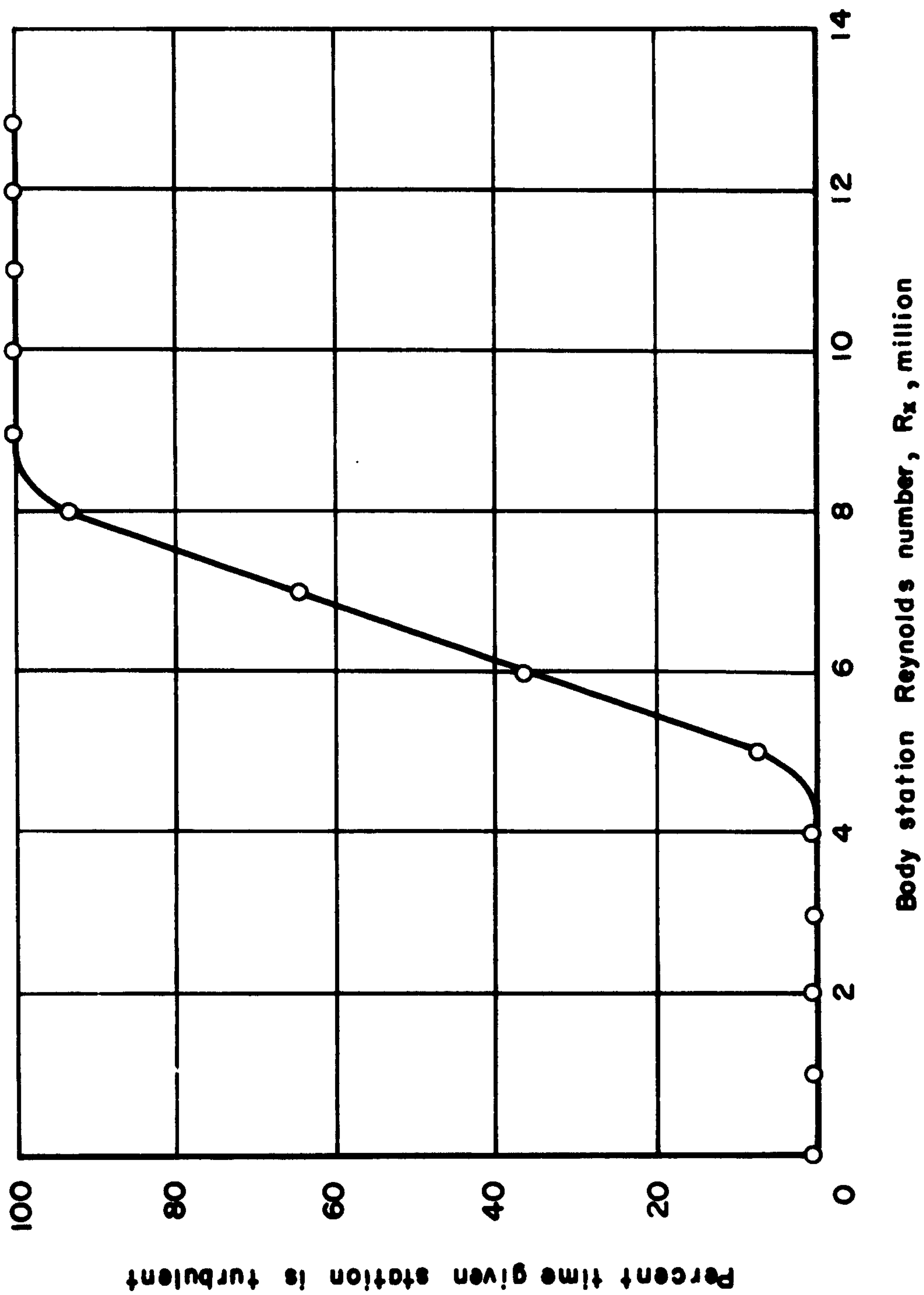


Figure 6.- Transition region for a 0.0015-inch screw-thread model, M = 4.95.

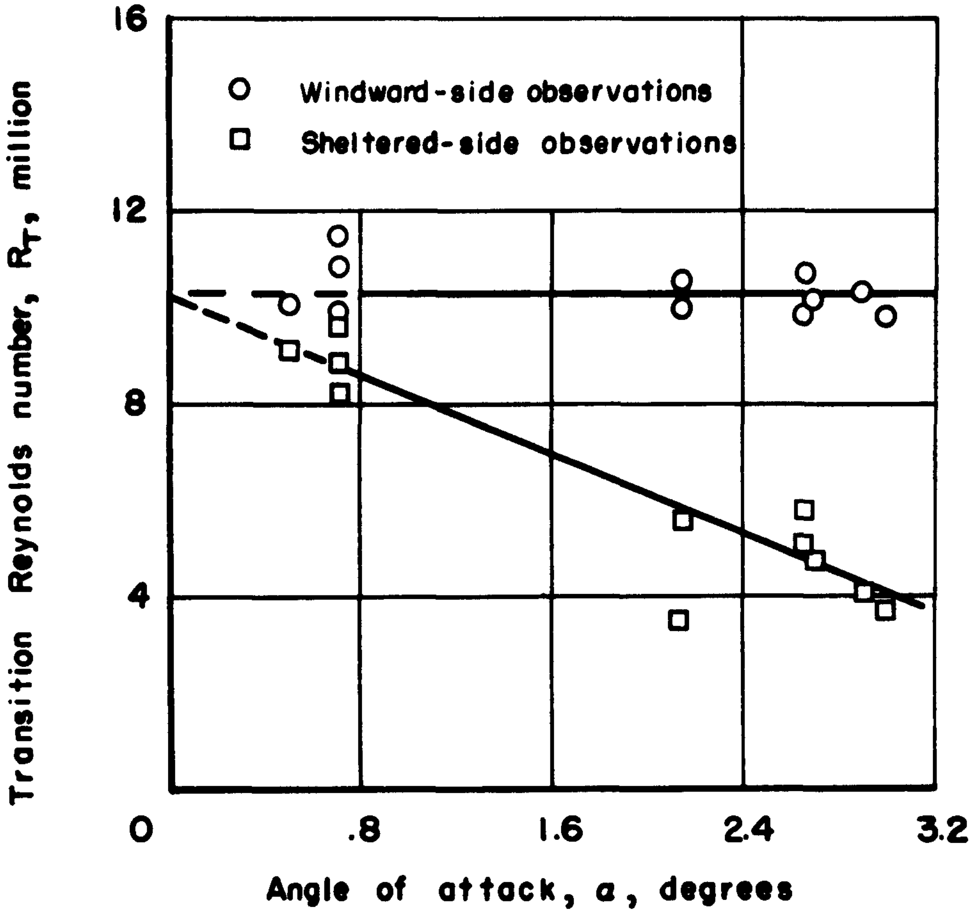


Figure 7.- Variation of windward-side and sheltered-side transition Reynolds number with angle of attack at M = 6.8.

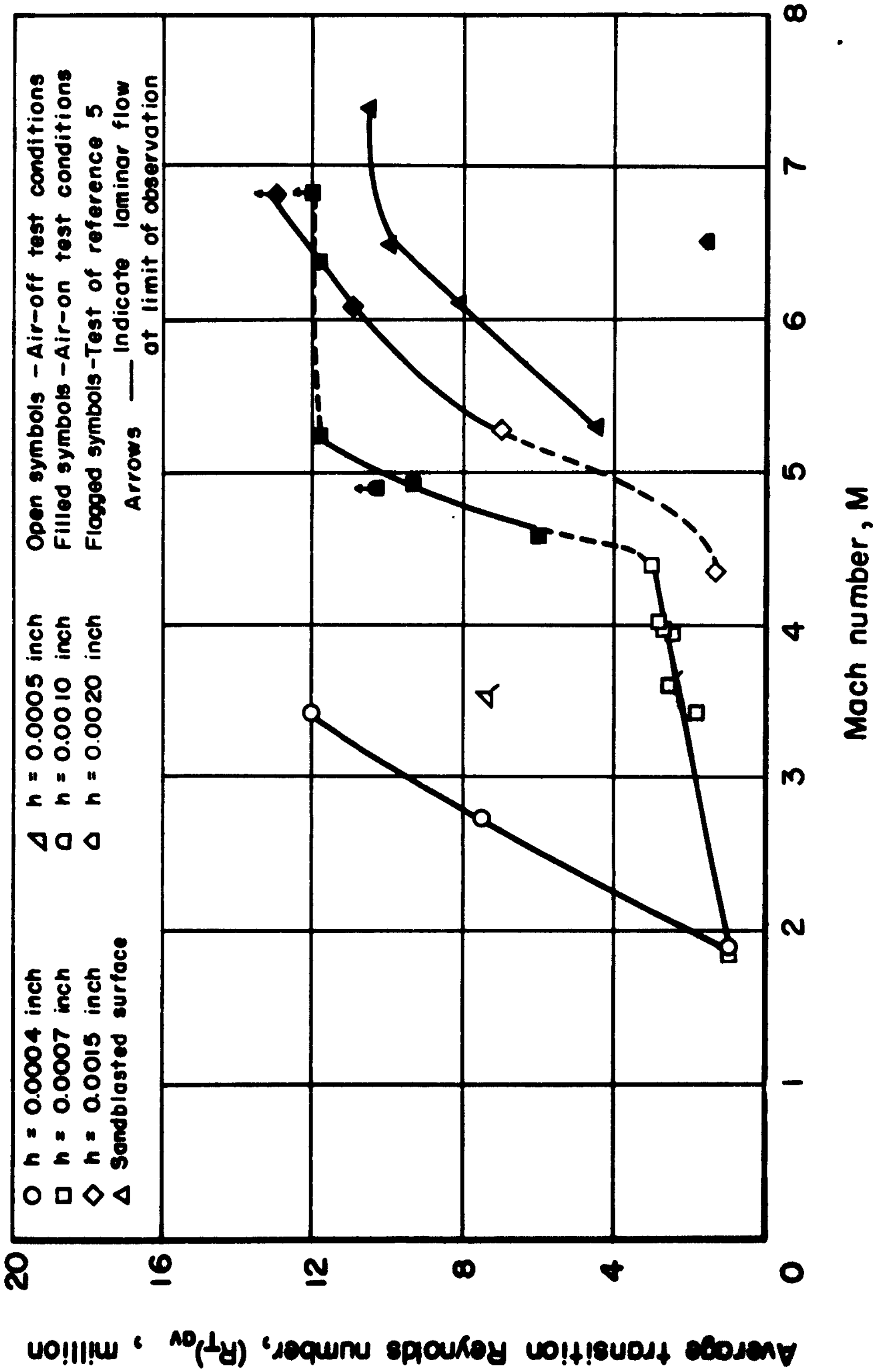


Figure 8.- Effect of Mach number variation on transition Reynolds number.

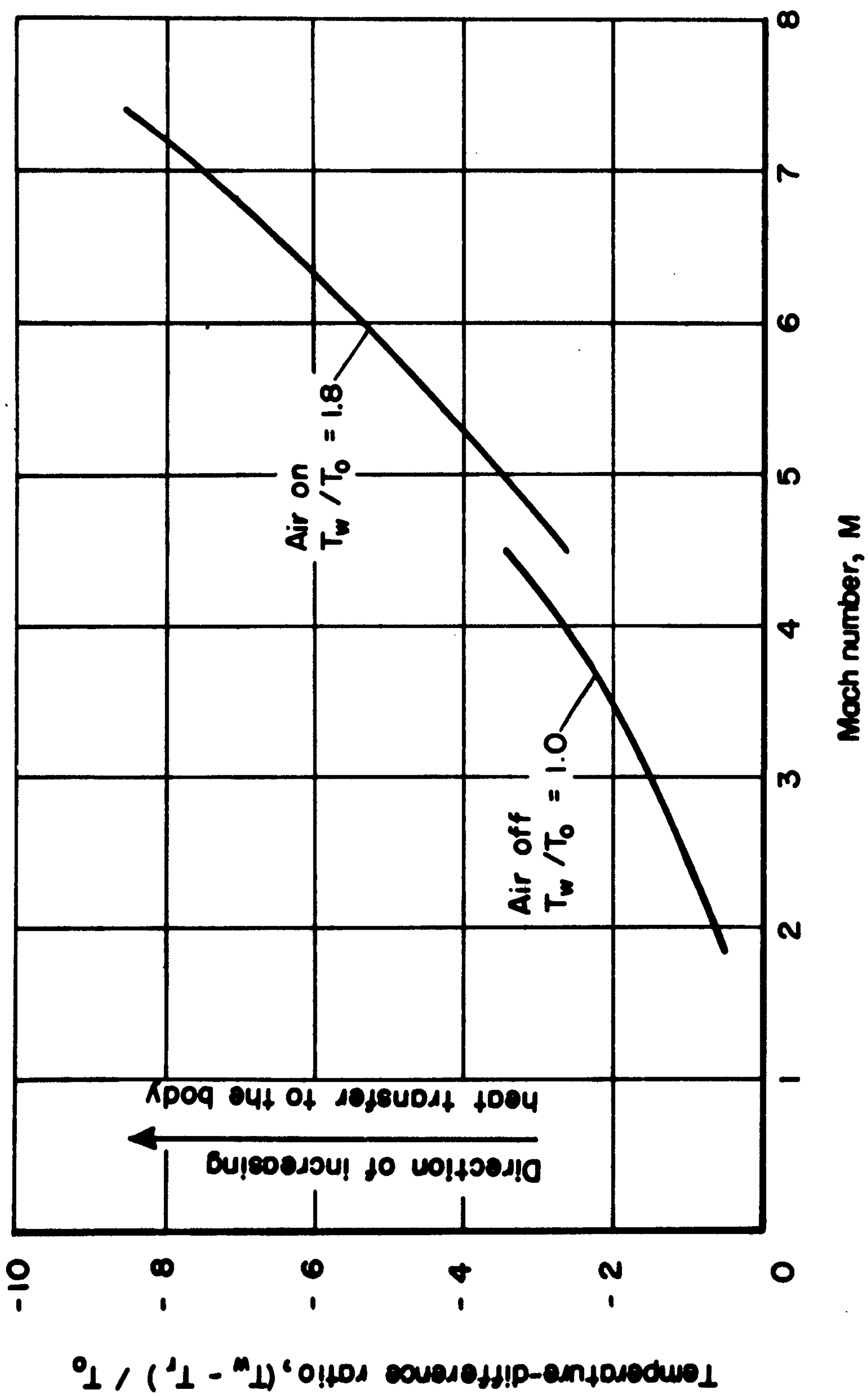


Figure 9.- Variation of temperature-difference ratio with Mach number for present test conditions.

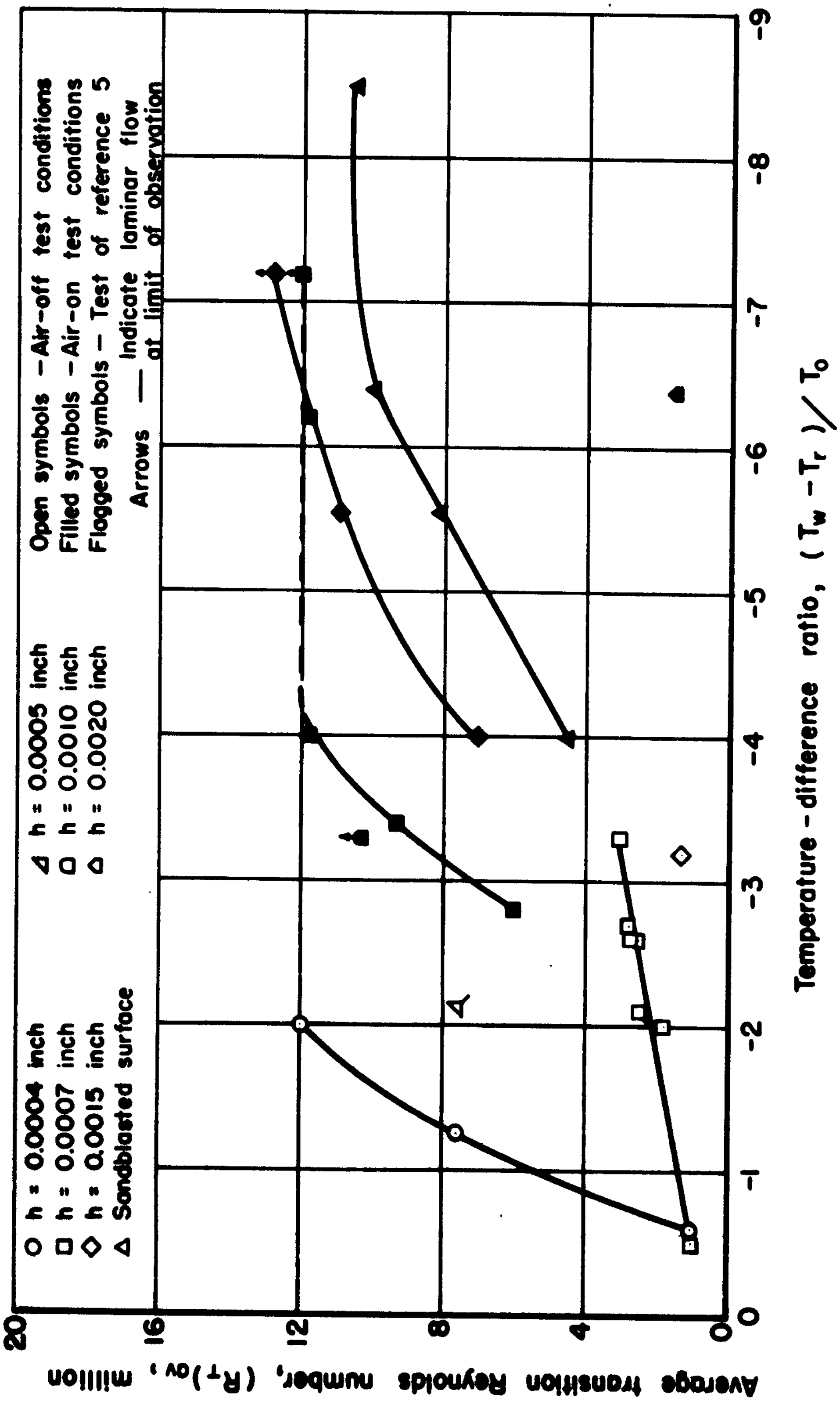


Figure 10.– Variation of transition Reynolds number with temperature-difference ratio for the Mach number range of the present test (1.8 to 7.4).

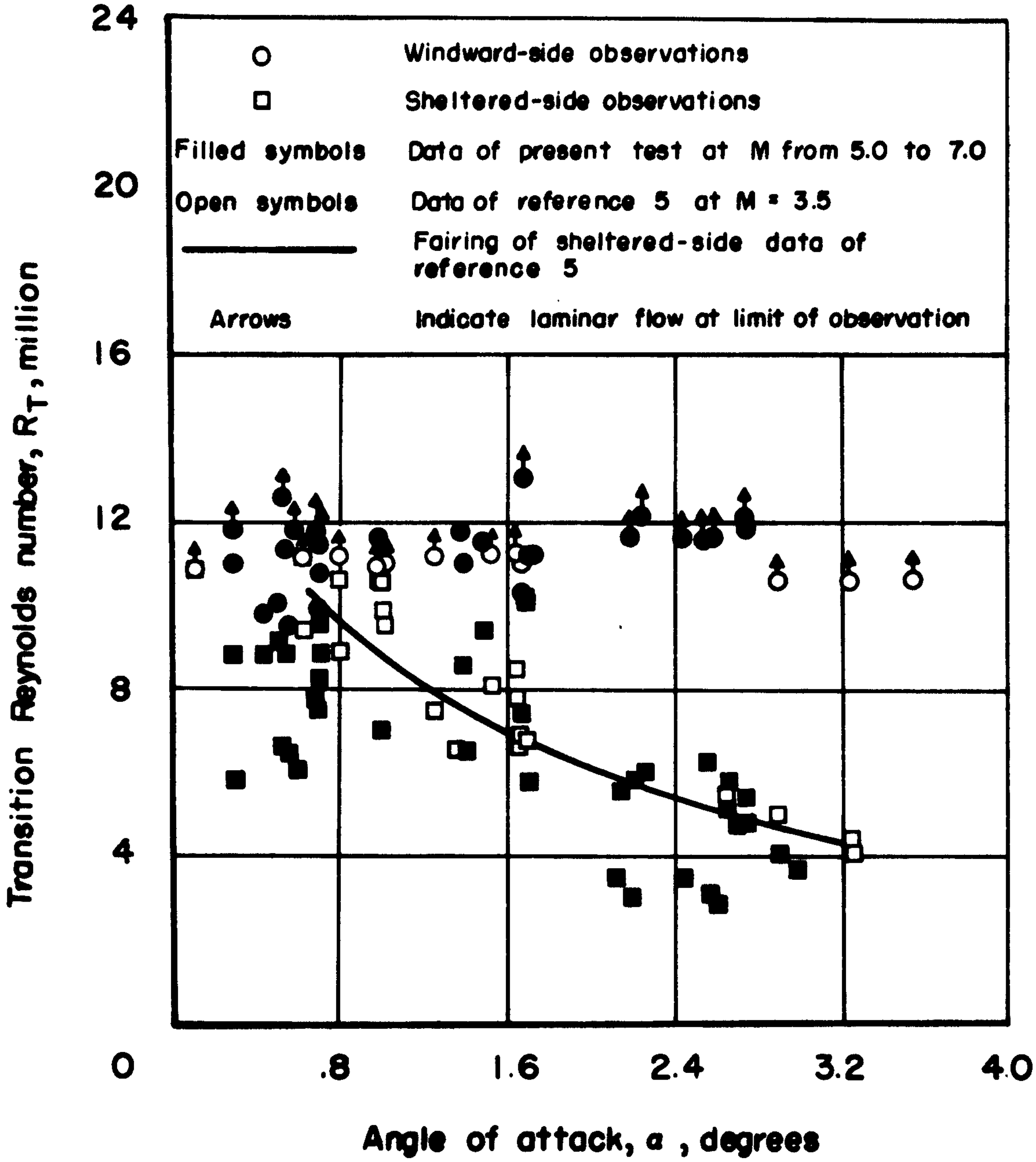


Figure 11.- Variation of windward-side and sheltered-side transition Reynolds number with angle of attack for the present test and reference 5.

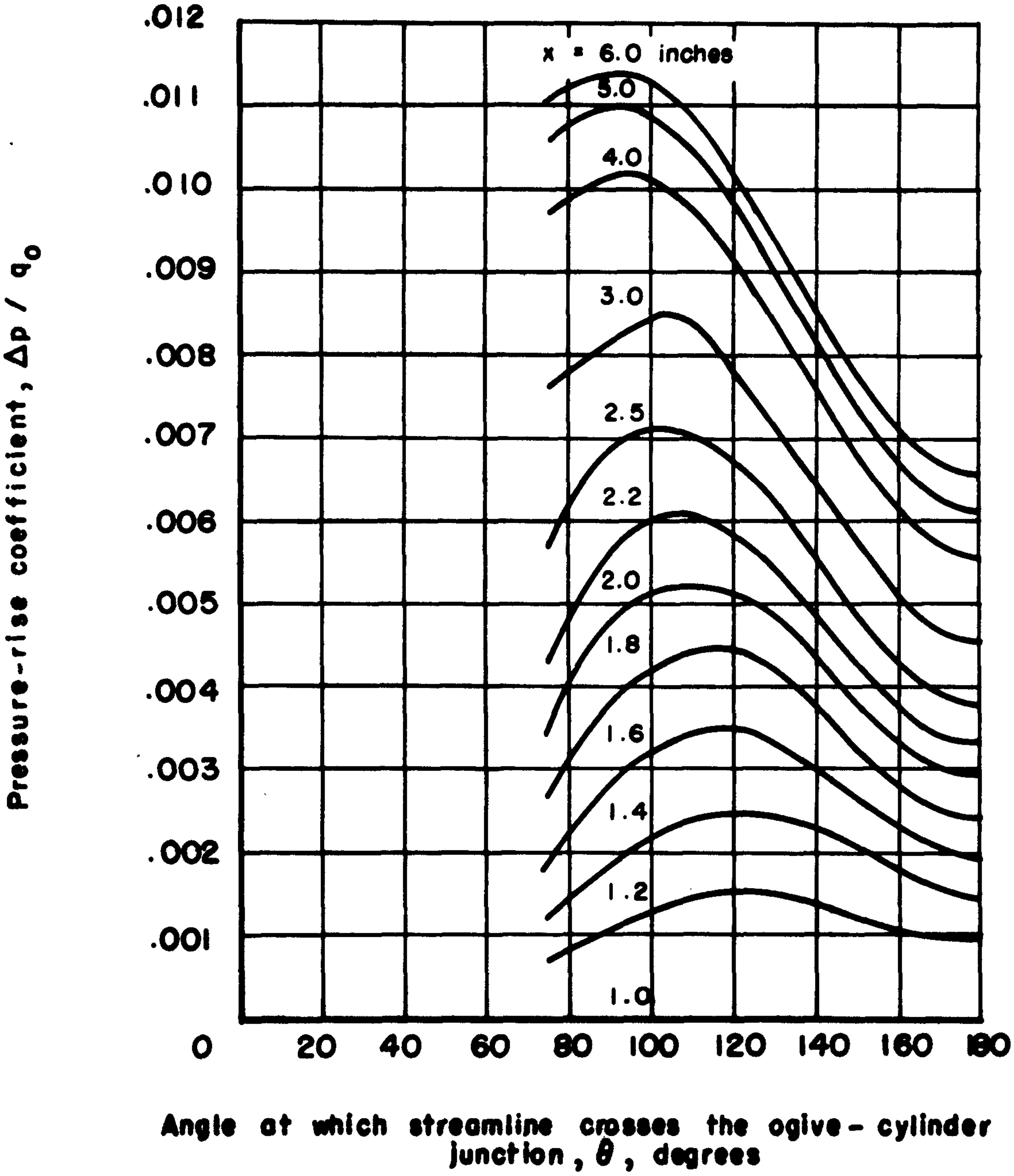


Figure 12.- Pressure rise along streamlines for several body stations; M = 6.8, α = 20°.

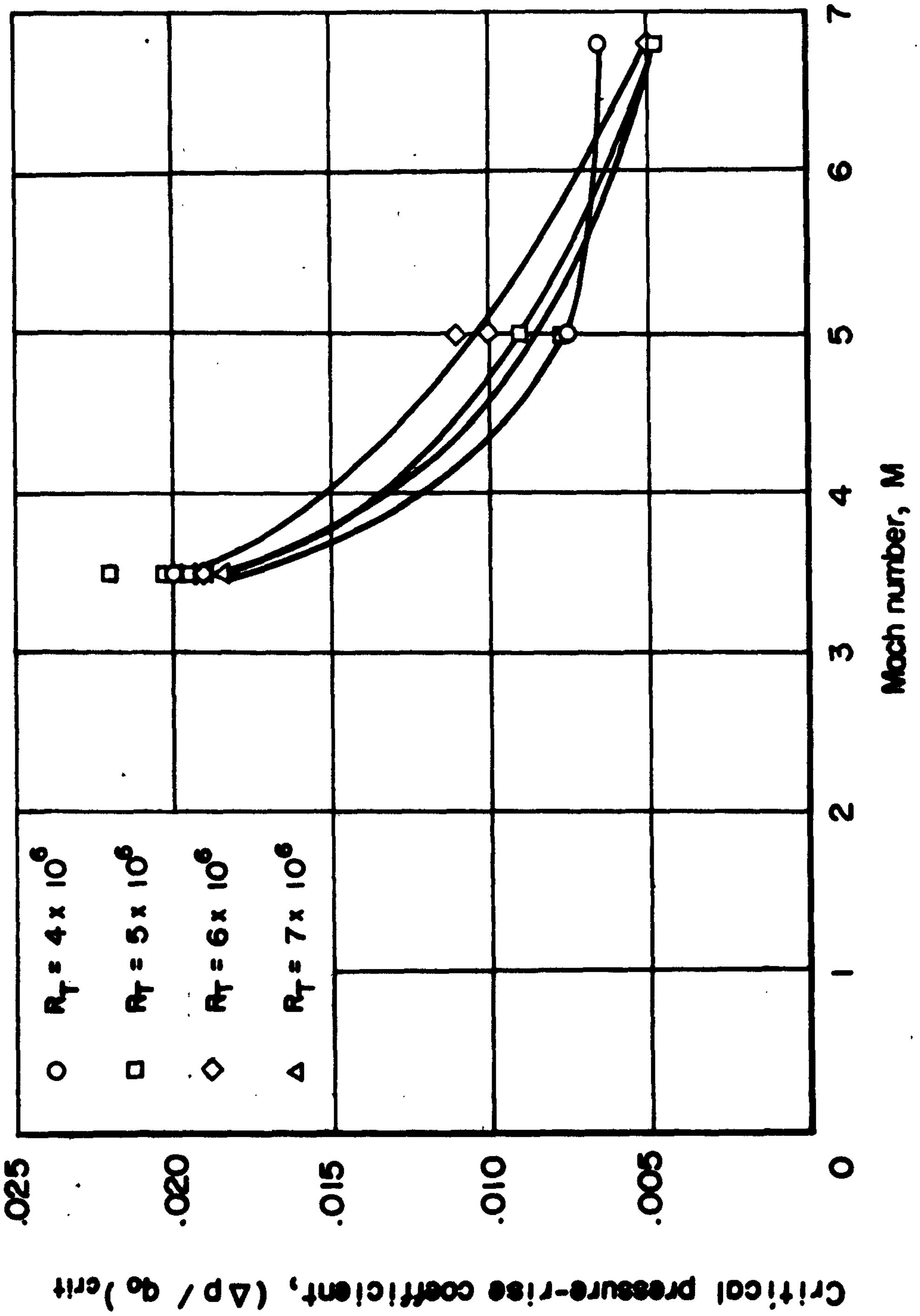


Figure 13.- Critical pressure-rise coefficient as a function of Mach number for several values of transition Reynolds number.

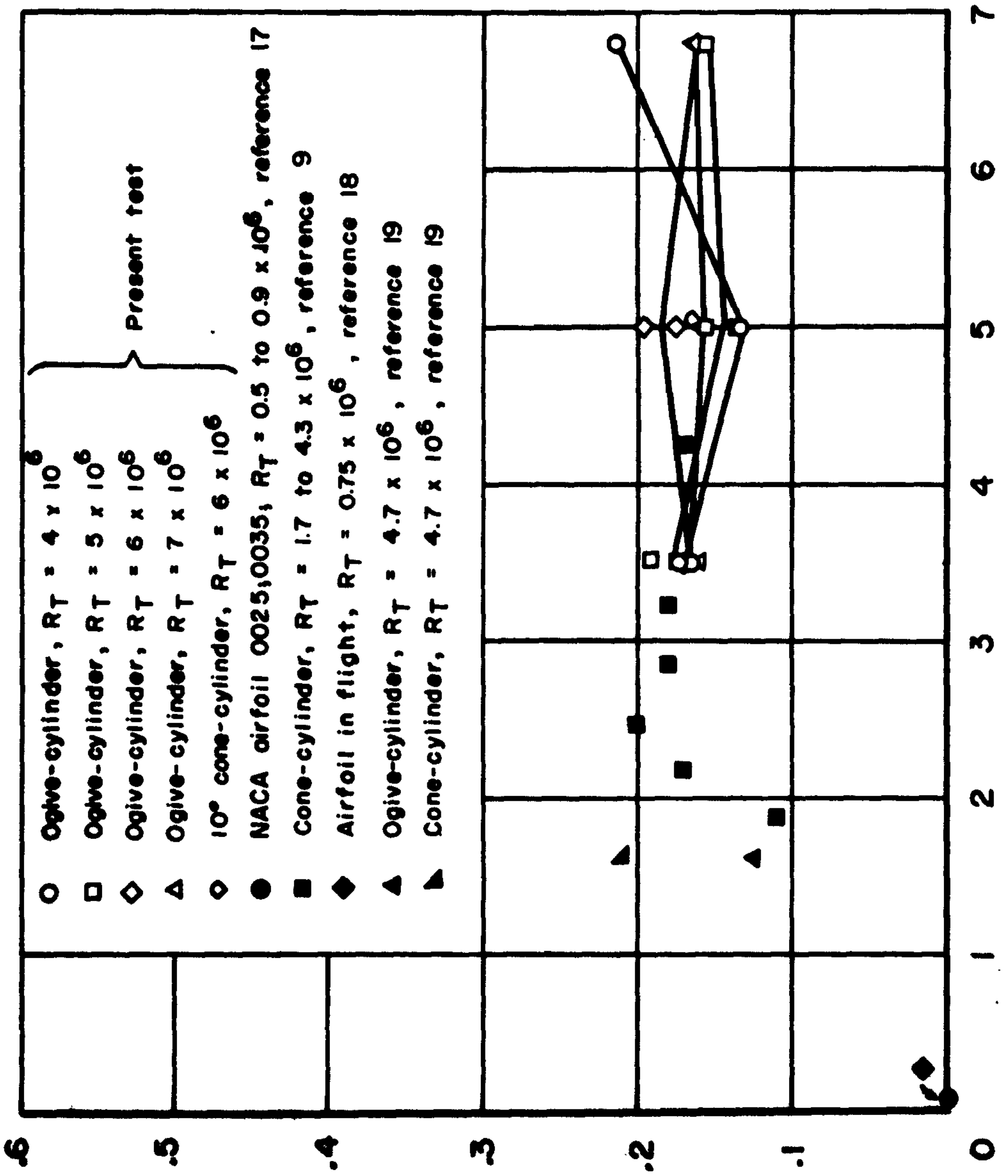


Figure 14.- Critical pressure-rise coefficient as a function of Mach number for a large variation in test conditions.

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